U.S. DEPARTMENT OF COMMERCE
National Technical Information Service

N78-27183

COMPOSITE STRUCTURES FOR COMMERCIAL TRANSPORT AIRCRAFT

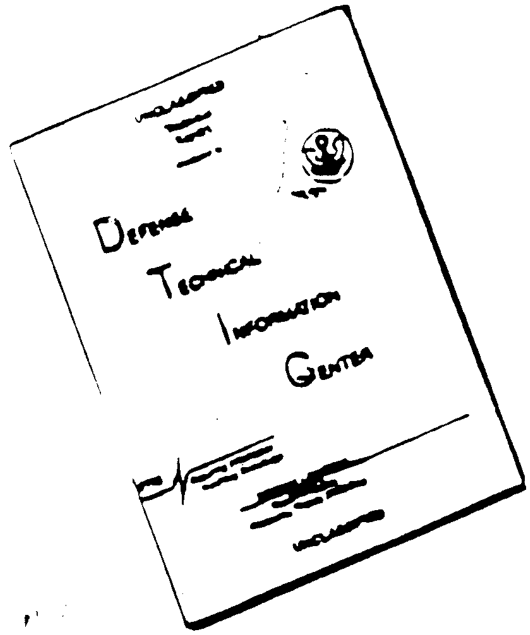
LOUIS F. VOSTEEN

NASA LANGLEY RESEARCH CENTER
HAMPTON, VIRGINIA

JUNE 1978

DEPARTMENT OF DEFENSE
PLASTICS TECHNICAL EVALUATION CENTER
ARRADCOM, DOVER, N. J. 07801

DISCLAIMER NOTICE



THIS DOCUMENT IS BEST QUALITY AVAILABLE. THE COPY FURNISHED TO DTIC CONTAINED A SIGNIFICANT NUMBER OF PAGES WHICH DO NOT REPRODUCE LEGIBLY.

NASA Technical Memorandum 78730

(NASA-TM-78730) COMPOSITE STRUCTURES FOR COMMERCIAL TRANSPORT AIRCRAFT (NASA) 25 p HC A02/MF A01 CSCL 11D

N78-27183

Unclas

G3/24 25217

COMPOSITE STRUCTURES FOR COMMERCIAL TRANSPORT AIRCRAFT

Louis F. Vosteen

June 1978

NASA

National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA TM 78730		

4. Title and Subtitle COMPOSITE STRUCTURES FOR COMMERCIAL TRANSPORT AIRCRAFT	5. Report Date June 1978 6. Performing Organization Code
7. Author(s) Louis F. Vosteen	8. Performing Organization Report No.
9. Performing Organization Name and Address NASA Langley Research Center Hampton, VA 23665	10. Work Unit No. 734-01 11. Contract or Grant No.
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546	13. Type of Report and Period Covered Technical Memorandum 14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

The Aircraft Energy Efficiency (ACEE) program was established to refine and demonstrate, in operational and production environments, technological improvements in engines, aerodynamics, electronics, and structures that could significantly improve the fuel efficiency of commercial transport aircraft. As part of the ACEE program, major development programs were instituted with the U.S. commercial transport builders, Boeing, Douglas, and Lockheed. The purpose of the program is to develop the technology and confidence that will enable the builders to commit to extensive use of composite structure on future transport aircraft. Six components, three secondary, and three primary structures, are presently under development. The six components are described along with the key features of the composite designs and their projected weight savings.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Composites Transport Aircraft Graphite-epoxy ACEE program	Unclassified - Unlimited STAR Category 24

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified		

* For sale by the National Technical Information Service, Springfield, Virginia 22161

COMPOSITE STRUCTURES FOR COMMERCIAL TRANSPORT AIRCRAFT

Louis F. Vosteen
Langley Research Center

SUMMARY

The development of graphite-epoxy composite structures for use on commercial transport aircraft is one of six major technology development efforts being conducted by the National Aeronautics and Space Administration (NASA) as part of its Aircraft Energy Efficiency (ACEE) program. Taken collectively, these six technologies have the potential for reducing the fuel consumption of commercial transports by up to 50 percent compared to today's jet transports.

Composite structures have a high potential for early application to transport aircraft, and the objective of the NASA program is to accelerate their development to the point where the commercial aircraft builders can incorporate these structures into their production aircraft. Six components, three secondary structures, and three primary structures, are presently under development. The six components are described along with some of the key features of the composite designs and their projected weight savings.

INTRODUCTION

In late 1975, the National Aeronautics and Space Administration (NASA) initiated an extensive program to improve the efficiency of current commercial transport aircraft through the development and application of some emerging technologies that, taken collectively, could reduce the fuel consumption of new aircraft by up to 50 percent in comparison to today's transports. This program, called the Aircraft Energy Efficiency (ACEE) Program is made up of six principal activities. These six activities are listed in Table I along with the percent fuel savings that each, taken alone, might contribute.

The objective of the ACEE program is to accelerate the development of these technologies to the point where commercial transport builders can incorporate the technology into their production aircraft. The time required to accomplish this will vary and depends on the current state of each technology and the rate at which the required additional development can proceed. Of the six technologies listed in Table I, composite structures are generally considered to have a high potential for early application to commercial transports. Additional information on overall program objectives and work content are described in reference 1.

TABLE I.- FUEL SAVING POTENTIAL OF TECHNOLOGIES INCLUDED IN THE AIRCRAFT ENERGY EFFICIENCY (ACEE) PROGRAM

o ENGINE COMPONENT IMPROVEMENT	5%
o ADVANCED "ENERGY EFFICIENT ENGINE"	10%
o ADVANCED TURBOPROPS	15-20%
o COMPOSITE STRUCTURES	10-15%
o AERODYNAMICS AND ACTIVE CONTROLS	10-20%
o LAMINAR FLOW CONTROL	20-40%
o COMBINED POTENTIAL	50%

This paper will describe the NASA-Industry program expected to lead to more extensive use of advanced composites. The term "advanced composites" generally refers to organic or metallic matrices reinforced with graphite, boron, or aramid fibers. In the remainder of this paper, these materials will be called simply composites.

Composite materials have been under development for more than 15 years. However, they have not been used extensively on commercial aircraft. A number of factors must be considered and certain information must be acquired before a new material or structural concept can be integrated into a production aircraft. Important considerations include a materials data base adequate for design; established, verified design procedures; experienced designers; demonstrated tangible benefits such as weight reduction or improved structural performance; assurance of durability, maintainability, and repairability; life-cycle costs comparable to or less than that of current materials and structures; adequate facilities; trained manufacturing personnel; and confidence that the product can be delivered in a timely way for a predictable and acceptable cost. Some of the factors can become barriers if sufficient information is not available. Some of the factors are largely technical while others are largely economic. Some technical concerns may become economic if the cost of solving them is prohibitive; and, conversely, what appears to be an economic concern might be resolved by the proper application of technology. The purpose of the ACEE composites program is to provide support to the transport aircraft industry that will develop the technology required.

The approach being taken in the ACEE program is to develop, for existing aircraft, components that have potential for significant weight savings, could, if economically practical, be integrated into current aircraft production, and serve as prototypes for similar classes of composite structure on future new aircraft.

COMPOSITE COMPONENTS

The components that have been selected for development are shown in figure 1 and their overall size and weight are shown in Table II.

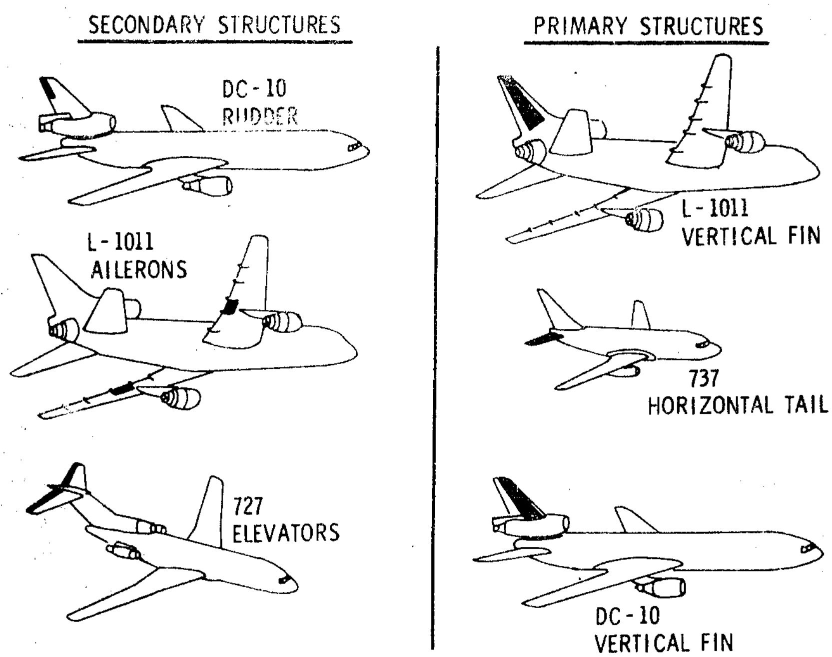


Figure 1.- Composite components being developed as part of ACEE program.

TABLE II.- SIZE AND WEIGHT OF ACEE COMPOSITE COMPONENTS

AIRCRAFT AND COMPONENT	COMPONENT DATA					
	PLANFORM AREA M^2	BASELINE WEIGHT kg	CHORD ROOT m	CHORD TIP m	SPAN m	COMPONENTS PER A/C
SECONDARY STRUCTURES						
727 ELEVATOR	4.1	128.3	1.24	0.53	5.26	2
DC-10 RUDDER	3.0	41.3	0.97	0.60	4.00	1
L-1011 AILERON	3.2	63.5	1.34	1.39	2.49	2
PRIMARY STRUCTURES						
737 HORIZ. STAB.	4.8	118.4	1.31	0.61	5.09	2
DC-10 VERT. FIN	9.36	423.4	2.07	1.10	6.95	1
L-1011 VERT. FIN	13.9	389.0	2.73	1.31	7.62	1

The selected components show potential for significant weight savings and offer an opportunity to develop design and manufacturing experience that could be applied to other aircraft components in the future. In all cases, the composite components are being designed to meet the same design requirements as the metal parts they replace. In addition, the parts are being designed to be interchangeable with the existing metal parts. For control surfaces, this means that existing hinge and actuator points are retained. Also, for both the secondary and primary components, the distribution and magnitude of loads being introduced into adjoining structure must be compatible with those structures in order to maintain interchangeability and avoid expensive modifications of existing structures. The design of each of the components will be described in subsequent sections of this paper.

DC-10 Rudder

The development of the DC-10 graphite-epoxy rudder began several years ago. The parts produced in the initial development program received FAA approval for use on the DC-10 and eight rudders have been installed. The results of the initial development programs are presented in references 2 and 3.

Figure 2 shows an exploded view of the upper aft rudder. This rudder segment is of multi-rib construction with two spars. Fiberglass leading and trailing edge members, a tip fairing, and aluminum hinge fittings complete the assembly. The structural box, consisting of the ribs and the two spars, is all graphite-epoxy and manufactured as a single, cocured unit.

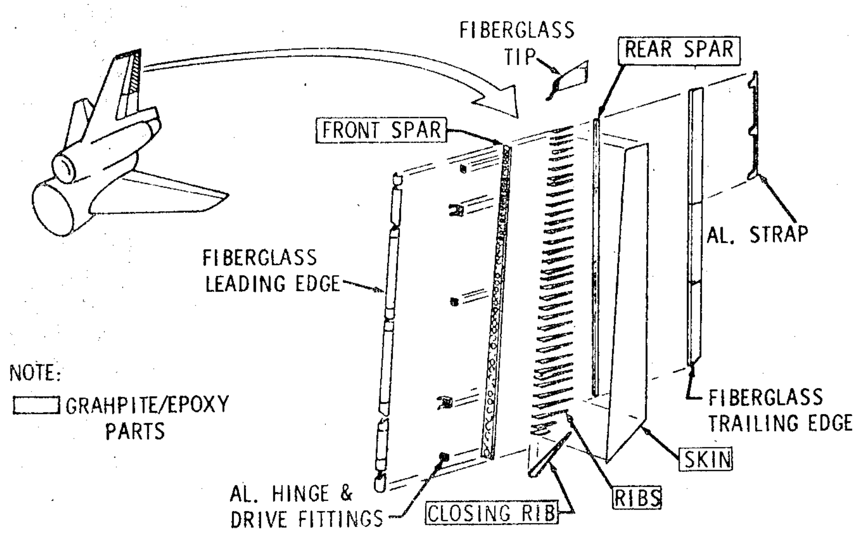


Figure 2.- Construction of graphite-epoxy DC-10 upper aft rudder.

Silicone rubber, which has a high coefficient of thermal expansion, is used within a steel mold tool to apply pressure to the graphite-epoxy material as the entire assembly is heated in an oven. Because the expanding rubber provides the pressure, an autoclave is not required. The first rudders were made by laying up 7.6 cm (3-inch) wide unidirectional tapes for both the substructure and the skins. In a follow-on effort, the contractor, Douglas Aircraft Company, investigated alternate material forms and fabrication techniques to try to further reduce the manufacturing cost of the graphite-epoxy rudder. The outcome of this study effort is shown in figure 3. Significant manhour savings are projected for the manufacture of additional rudders in the current program. These savings arise

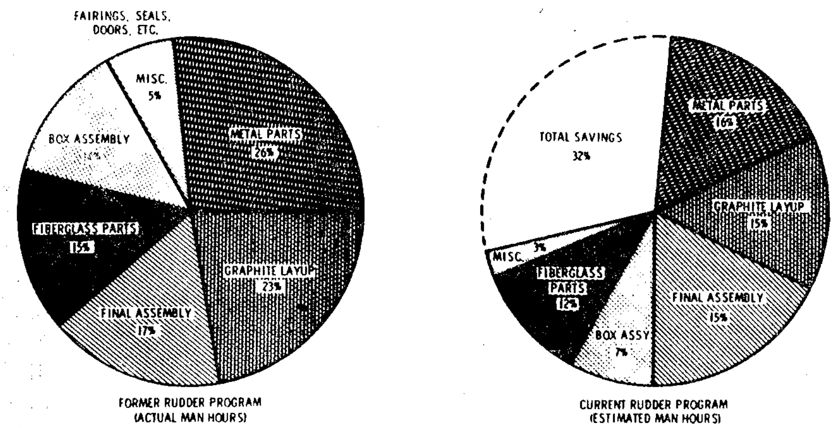


Figure 3.- Distribution of manufacturing labor for prior rudder program and estimate of labor savings for revised manufacturing process.

from improvements in the assembly tooling and manufacture of common parts as well as improvements in the layup and fabrication of the graphite parts. The reduction in graphite layup time accounts for about one-third of the estimated labor savings and results primarily from a change from narrow, unidirectional tape to broadgoods and woven fabric. Unidirectional broadgoods are being used for skins and woven fabrics are being used for ribs and spars.

The graphite-epoxy upper aft rudder is 30 percent lighter than its aluminum counterpart and initial cost projections indicate a cost comparable to the aluminum part once production is underway. Because the rudder is

a mass balanced part, further weight savings could be achieved by reducing the balance weights. However, the composite rudder is interchangeable with the aluminum rudder, and therefore, the balance weights (which are forward of the hinge line of the forward rudder) would have to be replaced if an aluminum rudder was ever installed to replace a composite rudder. Since extra balance weight is acceptable from aeroelastic considerations, the contractor has chosen to avoid this complication at the present time but would probably incorporate such a change when composite rudders become production items.

Ten rudders will be built during this year to obtain quantitative cost data.

L-1011 Inboard Aileron

The inboard aileron of the L-1011 is located aft of the wing-mounted engines. Design concepts for the aileron were investigated by the Lockheed-California Company in an earlier NASA contract and the results of the study are given in references 4 and 5. As part of the ACEE program, a new contract has been initiated with Lockheed for the composite aileron development.

The design of the present aluminum aileron is shown in figure 4. Several designs have been considered for the composite aileron. One design has been selected for further analysis and is also shown in figure 4.

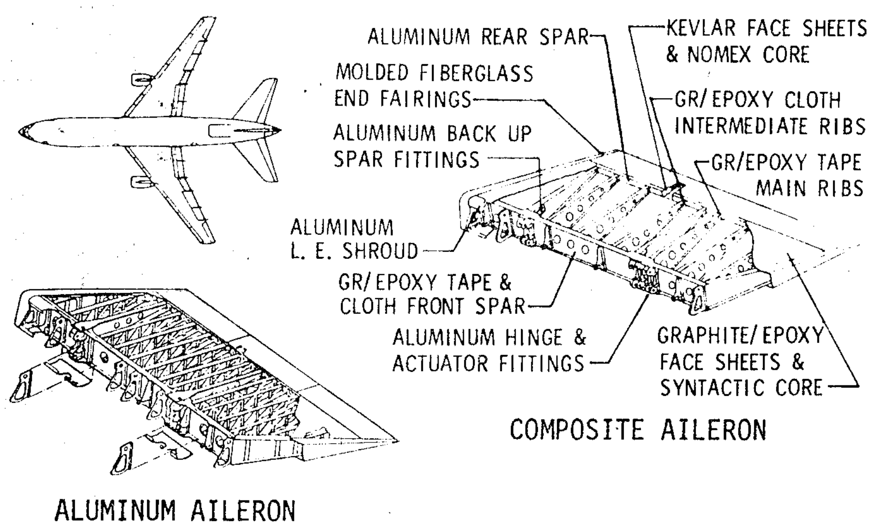


Figure 4.- Comparison of current L-1011 aluminum aileron design and selected composite design.

The composite design features a thin syntactic-foam-core sandwich with graphite-epoxy face sheets. The internal structure is primarily graphite-epoxy and the number of intermediate ribs has been reduced from 13 in the aluminum design to 5 in the composite design. The syntactic-foam sandwich appears to be more resistant to impact damage than a honeycomb-core sandwich. The impact damage environment the aileron must be able to withstand comes from hailstone impact on the ground. Some preliminary tests show that the syntactic foam sandwich covers will sustain no detectable damage at impact energies well above that produced by an 18-mm diameter hailstone falling at terminal velocity.

The program has been underway for only a short time, so little quantitative data are available.

Preliminary design work indicates the potential for about 25 percent weight saving in relation to the metal aileron and a substantial reduction in the number of separate parts and fasteners that will be needed in the final assembly. Part reduction and fastener reduction are two important factors that can make composite structure more economical to assemble than conventional aluminum structure.

Ten shipsets of ailerons (10 right hand and 10 left hand) will be fabricated to establish a basis for projecting production costs.

727 Elevators

A graphite-epoxy elevator is being developed by the Boeing Commercial Airplane Company under a NASA contract. The design makes extensive use of honeycomb sandwich structure for both the surface panels and interior ribs. A comparison of the graphite-epoxy elevator with the present aluminum design is shown on figure 5. The efficiency of sandwich construction for carrying normal pressure loads and in-plane shear permit a design in which most of the interior ribs are removed. Over most of the surface, the honeycomb face sheets are two plies thick. One ply is unidirectional tape and the other is woven fabric. The tape is used as the outer ply on the outer skin because it gives a better surface for finishing. On the inner face, the tape is next to the core and fabric is used on the exposed surface. Fabric is somewhat more resistant to fiber breakout during drilling and, therefore, is used on the exit side of drilled holes.

A number of major subcomponent tests are planned to verify the design. An outline of the subcomponents, showing their locations on the elevator, can be seen on figure 6. The elevator lies in a strong acoustic field, therefore, some of the early tests will evaluate the response of the large unsupported skin panels to acoustic loads and the adequacy of the skin to spar attachments. The design of the elevator is dictated primarily by stiffness requirements, so the strain levels tend to be low. The calculated strain at ultimate load is generally less than 0.002 with a maximum of 0.004 in a limited region near the actuator rib.

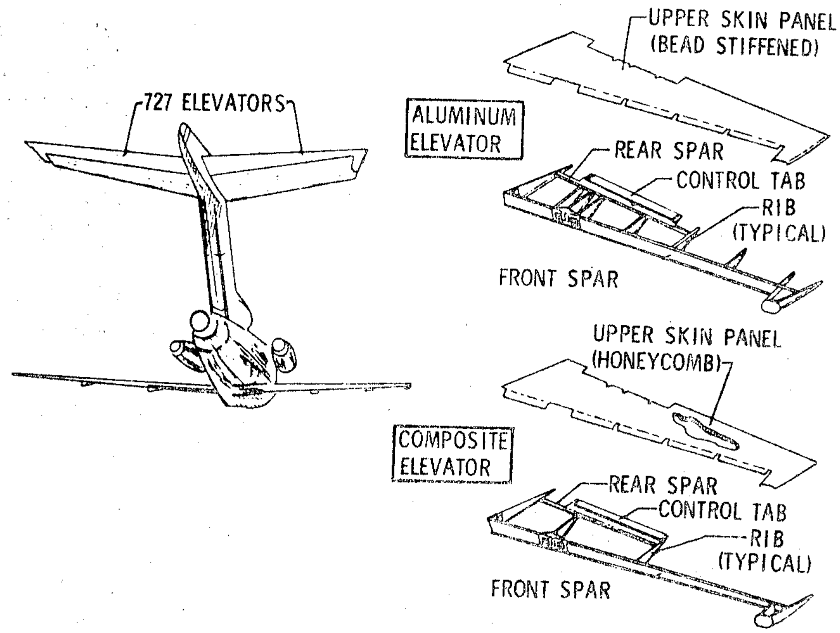


Figure 5.- Comparison of current 727 aluminum elevator design and selected composite design.

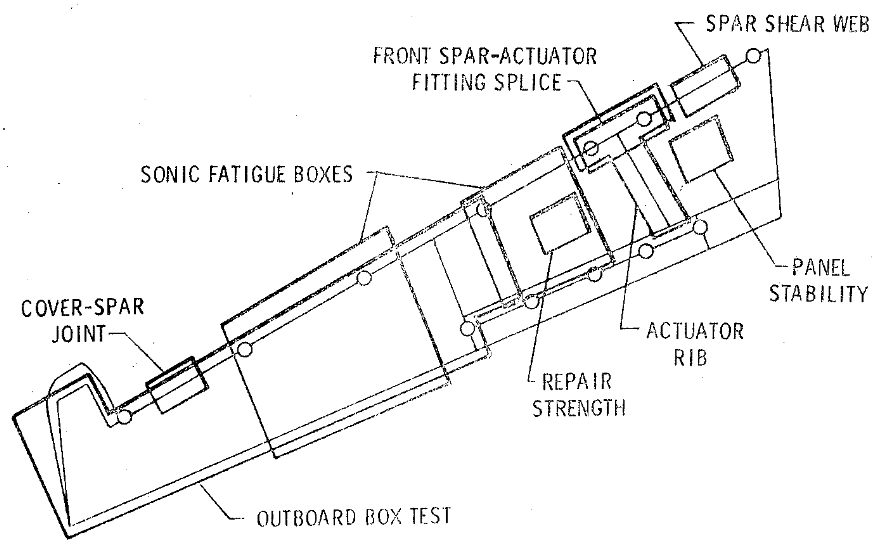


Figure 6.- Major design verification test components for 727 elevator.

The composite portion of the elevator weighs 22 percent less than the aluminum part. (See Table III.) The elevator is also mass-balanced, and a

TABLE III.- COMPARISON OF WEIGHT DISTRIBUTIONS FOR ALUMINUM AND COMPOSITE ELEVATOR DESIGNS

	ALUMINUM DESIGN WT. kg	COMPOSITE DESIGN WT. kg	WEIGHT CHANGE kg	PERCENT CHANGE
FRONT AND REAR SPARS	35.2	26.3	- 8.9	- 25
RIBS	12.1	7.1	- 5.0	- 41
SKIN PANELS	52.7	44.5	- 8.2	- 16
CONTROL TAB	11.1	6.1	- 5.0	- 45
HORN STRUCTURE	6.0	3.6	- 2.4	- 39
CORROSION PROVISIONS	0	3.0	+ 3.0	-
LIGHTNING STRIKE PROVISIONS	0	1.2	+ 1.2	-
TOTAL REPLACED STRUCTURE (METAL TO COMPOSITE)	117.1	91.8	-25.3	- 22
BALANCE PANEL WEIGHTS	32.0	0	-32.0	-100
BALANCE PANEL HINGES	54.6	44.9	- 9.7	- 18
TOTAL REVISED STRUCTURE	86.6	44.9	-41.7	- 48
NOSE RIBS AND SKINS	18.0	18.0	0	0
BALANCE PANEL STRUCTURE	16.0	16.0	0	0
HORN BALANCE WEIGHT	18.8	18.8	0	0
TOTAL COMMON STRUCTURE	52.8	52.8	0	0
TOTAL FOR AIRPLANE SYSTEM (TWO ELEVATORS)	256.5	189.5	-67.0	- 26

further weight saving can be effected. The amount of balance weight that can be removed actually exceeds the weight saving on the elevator box because of the difference between the effective lever arms. In this case, the contractor will take advantage of the weight saving on the balance weights because the weights can be readily replaced along with the elevator. An overall weight saving of 26 percent is estimated.

L-1011 Vertical Tail

The overall configuration of the L-1011 vertical tail is shown in figure 7. The aluminum design has 2 main spars and 17 ribs. Load is introduced into the fuselage at the spars and through a continuous skin splice joint. The composite design retains the same basic 2-spar configuration, but the number of ribs has been reduced to 12. The estimated weight saving in the new design is 28 percent.

Epoxy resins are known to absorb moisture and both moisture and temperature affect the strength of graphite-epoxy composites. Figure 8 shows the tensile and compressive strengths of notched cross-plied laminates as a

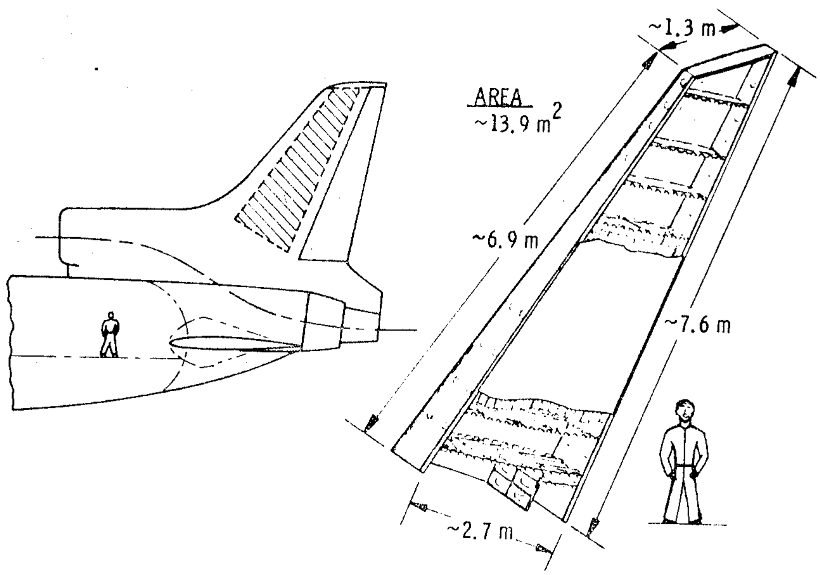


Figure 7.- L-1011 vertical tail configuration.

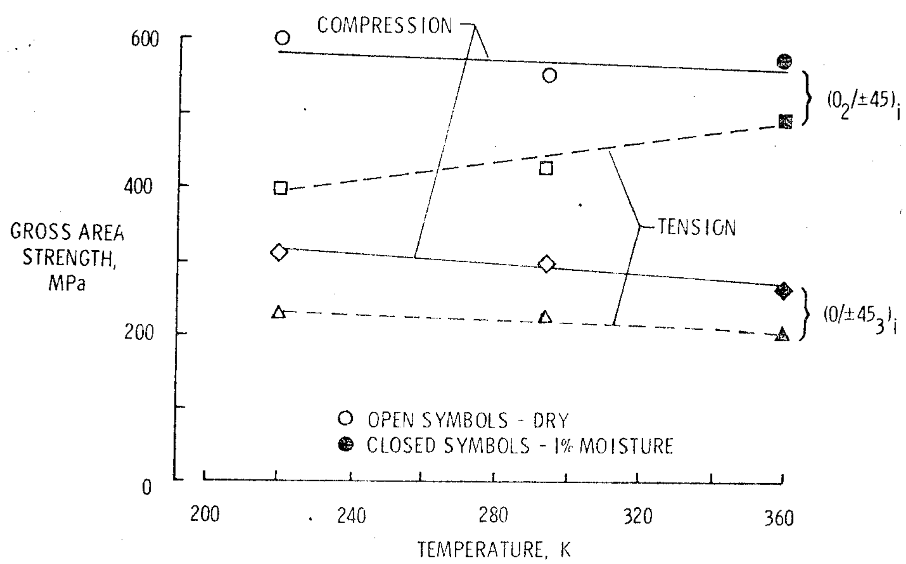


Figure 8.- Effect of temperature on the strength of notched laminates (Narmco 5208/T300).

function of temperature over the range of temperature of particular interest for commercial aircraft. The data were obtained by the Lockheed-California Company under the present contract. The data at 220K and 295K are for "dry" laminates. The laminates tested at 360K had 1 percent moisture by weight. The lower temperatures can be reached at high altitude on a cold day. The higher temperature would occur only rarely, but represents an extreme design condition. As shown on the figure, moisture and elevated temperature have only a small effect on the properties over this temperature range. Additional data on behavior of this material (Narmco 5208/T300) is given in reference 6.

Both woven fabric and unidirectional tape material have been considered for the skin plies. Unidirectional tape was selected because it offered slightly better weight savings, and, more importantly, it has a higher potential for manufacturing cost savings because tape is more adaptable to automated layup processes. Automated layup is particularly advantageous for this part because the large surface area and physical dimension of the skin complicate hand layup.

The present skin design uses hat-shaped stiffeners cocured with the skin layup. A process being investigated to form the hats is a roll-forming process not unlike that used to form metal parts. A schematic of the process is shown in figure 9. Layers of material, some of them

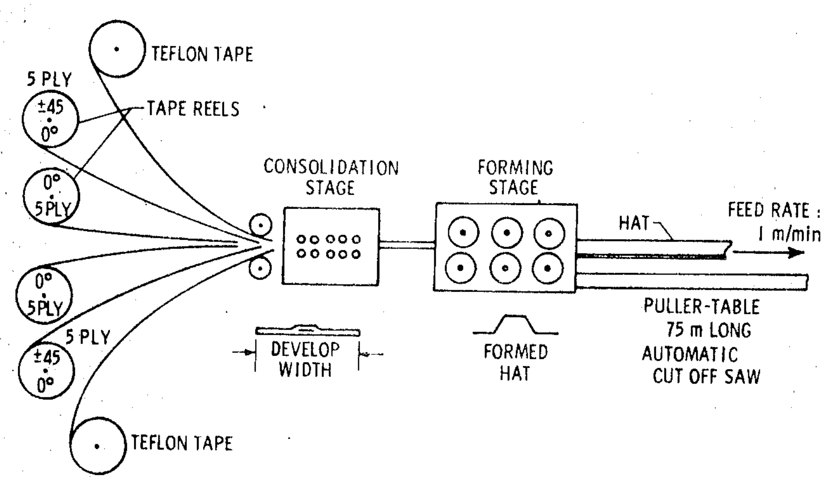
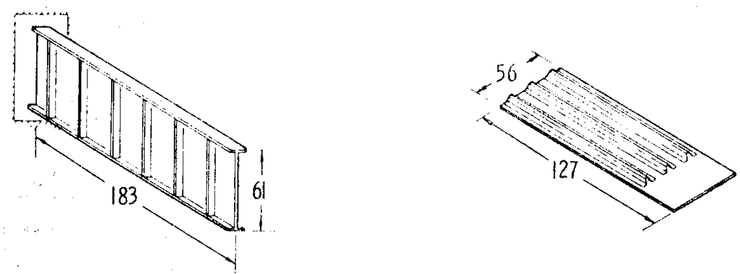


Figure 9.- Schematic diagram of proposed roll-form process for producing composite hat section stiffeners.

preplied to facilitate handling, are passed through heated rollers that align the layers and provide initial compaction. This "flat stock" then passes through the forming rolls and is partially cured to assure shape retention. The preformed hats are placed on the skin layup and the assembly is cocured in an autoclave.

The long-term durability of composites in the operating environment is still a question of concern to the manufacturers. As part of the tail development program, a series of long-term cyclic load environmental tests will be performed. The component configurations are shown on figure 10 and represent a spar segment near the root attachment and a cover panel



(a) SPAR SEGMENT		(b) COVER SEGMENT
10	DURABILITY SPECIMENS	10
10	STATIC STRENGTH SPECIMENS	10
20		20

Figure 10.- Spar and skin panel components to be used in long-term durability test program (all dimensions are in centimeters)

including the skin root joint. Twenty components of each will be built. One purpose of the program is to better define the types of manufacturing defects that might be expected in production. The components will be built in production facilities on production tools. Parts will be carefully inspected to locate possible defects. Some defects may require repair; others will be allowed to remain and their rate of growth will be monitored during the tests. A typical test cycle is shown in figure 11. Temperature, humidity, and load will be varied. The ground-air-ground cycle will be simulated in near real time, but loads will be applied in blocks of about 40 cycles spectrum load in each flight phase. Temperature extremes will not occur on each flight, but will still be more severe than those expected in normal service. The tests will run for sufficient time to simulate a 20-year service life. Other major test components, and the regions of the fin which they will represent, are shown in figure 12.

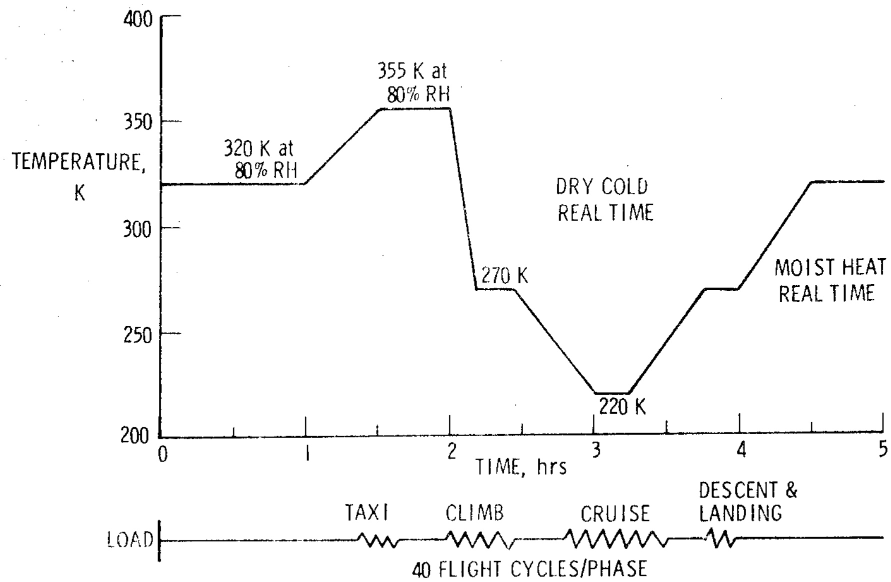


Figure 11.- Typical test cycle for long-term durability tests.

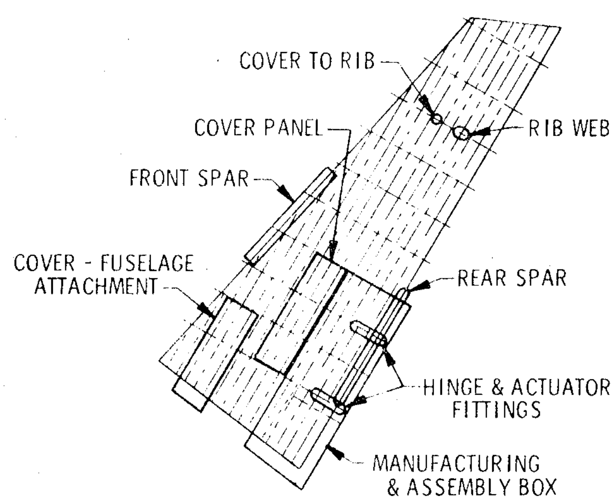


Figure 12.- Major design verification test components
for L-1011 vertical tail.

ORIGINAL PAGE IS
OF POOR QUALITY

Early in the program, a manufacturing development component was made to check the manufacturing and assembly processes. A photograph of the completed box is shown in figure 13. The box was about 3-m long and 1.5-m wide. A comparison of the weight of the original aluminum tail and the current design weight of the composite tail is shown in Table IV.

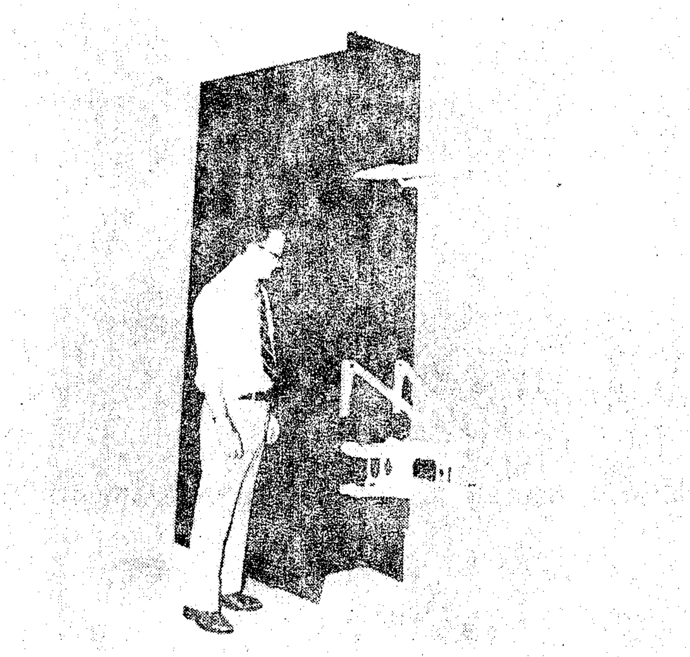


Figure 13.- L-1011 vertical tail manufacturing development component.

TABLE IV.- COMPARISON OF WEIGHT DISTRIBUTIONS FOR ALUMINUM AND COMPOSITE L-1011 VERTICAL TAIL

	ALUMINUM DESIGN WT. kg	COMPOSITE DESIGN WT. kg	WEIGHT CHANGE kg	PERCENT CHANGE
FRONT AND REAR SPARS	90.3	55.2	- 35.1	- 39
RIBS	69.5	48.7	- 20.8	- 30
COVERS	208.8	160.4	- 48.4	- 23
ASSEMBLY HARDWARE	16.1	6.5	- 9.5	- 59
PROTECTIVE FINISH	4.3	4.3	0	0
LIGHTNING PROTECTION		6.4	+ 6.4	—
TOTAL WEIGHT	389.0	281.5	-107.5	- 27.6

737 Horizontal Tail

A composite horizontal tail is being developed by the Boeing Commercial Airplane Company for the model 737 aircraft. The present aluminum design, figure 14, is a 2-spar, multi-rib box with unstiffened skins.

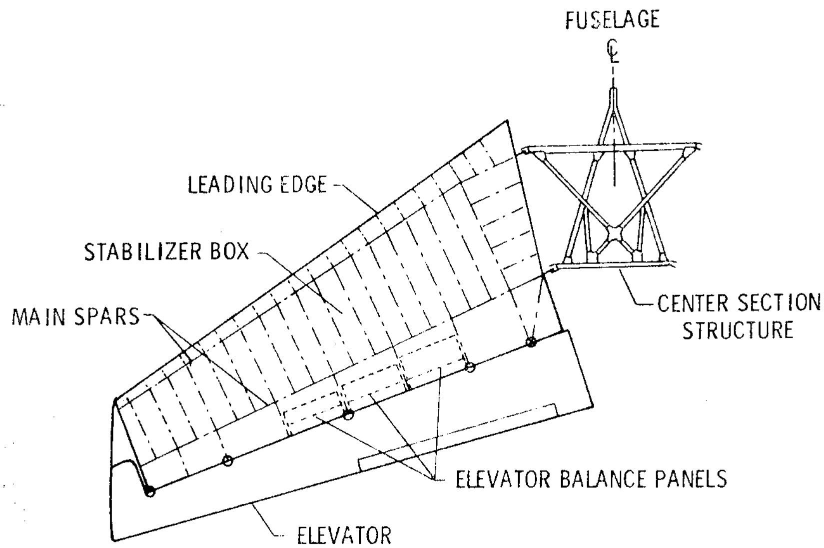


Figure 14.- General arrangement of aluminum 737 horizontal tail.

The box attaches to the fuselage structure through 5 lugs, 2 on the front spar and 3 on the rear. The composite design, shown in figure 15, retains the same fuselage attach points, but about two-thirds of the ribs have been removed and an integrally stiffened skin is used. The inspar ribs will have honeycomb sandwich webs.

Major load transfer points pose special design problems for composites. In the horizontal stabilizer, all the load must be transferred to the lug attachments. The composite stabilizer will use titanium reinforcement straps that are bonded and bolted to built-up spar chords. A sketch of the reinforced lug and the buildup of the spar chords is shown in figure 16. The heavy chord sections will be made by bonding precured chord elements into the spar assembly during its cure.

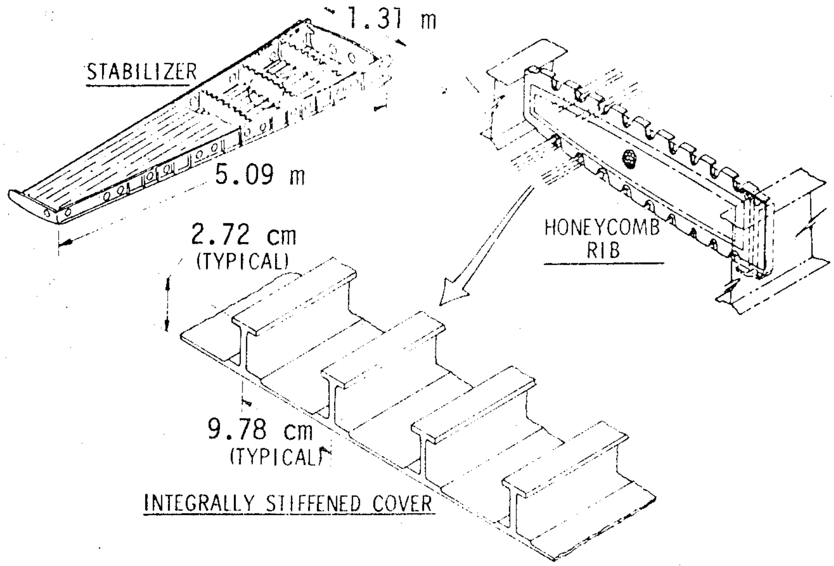


Figure 15.- Design features of composite 737 horizontal tail.

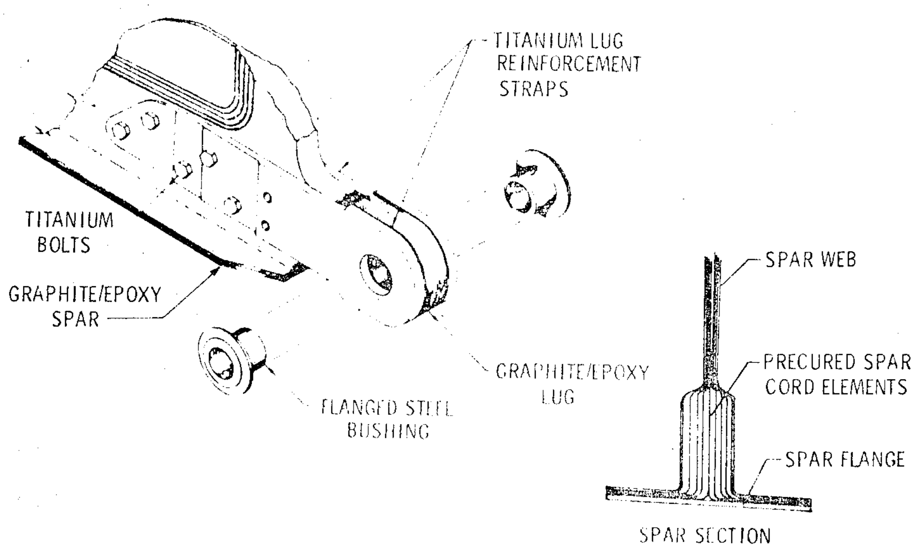


Figure 16.- Titanium reinforced attachment lug for 737 horizontal tail.

The weight of the composite stabilizer compared to the aluminum stabilizer is shown in Table V. Major weight savings are in the ribs and spars. The composite skins are only slightly lighter than the aluminum skins, but because of their increased stiffness, the rib spacing, and, in turn, the number of ribs are reduced substantially.

Five shipsets of stabilizers will be constructed to establish manufacturing costs.

TABLE V.- COMPARISON OF WEIGHT DISTRIBUTIONS FOR ALUMINUM AND COMPOSITE 737 HORIZONTAL TAIL

	ALUMINUM DESIGN WT. kg	COMPOSITE DESIGN WT. kg	WEIGHT CHANGE kg	PERCENT CHANGE
FRONT SPAR	31.3	20.2	- 11.1	- 35
REAR SPAR	71.1	42.9	- 28.2	- 40
RIBS	60.9	30.3	- 30.6	- 50
COVERS	72.4	64.6	- 7.8	- 11
CORROSION PROTECTION	---	6.8	+ 6.8	—
LIGHTNING PROTECTION	---	0.4	+ 0.4	—
ACCESS DOORS	0.7	0.9	+ 0.2	+ 31
TOTAL WEIGHT (INCLUDES LH & RH STABILIZERS)	236.4	166.1	- 70.3	- 29.7

DC-10 Vertical Stabilizer

The size and general arrangement of the DC-10 vertical stabilizer is shown on figure 17. The stabilizer is attached to the fuselage through 8 tension bolts, 2 on each spar. As they were for the 737 horizontal stabilizer, these major load transfer points are an important part of the composite design. The present design is a stepped titanium fitting bonded into the spar during cure. Bolts will be added to reduce the peel loads on the bond. A sketch of this arrangement is shown in figure 18.

The skin of the stabilizer is lightly loaded and will be honeycomb sandwich construction. An integrally stiffened skin was also considered for this design, but sandwich skins appear to have manufacturing advantages. In addition, some of the spar cap material will be integrated with the skin by replacing the honeycomb over the spars by solid laminate.

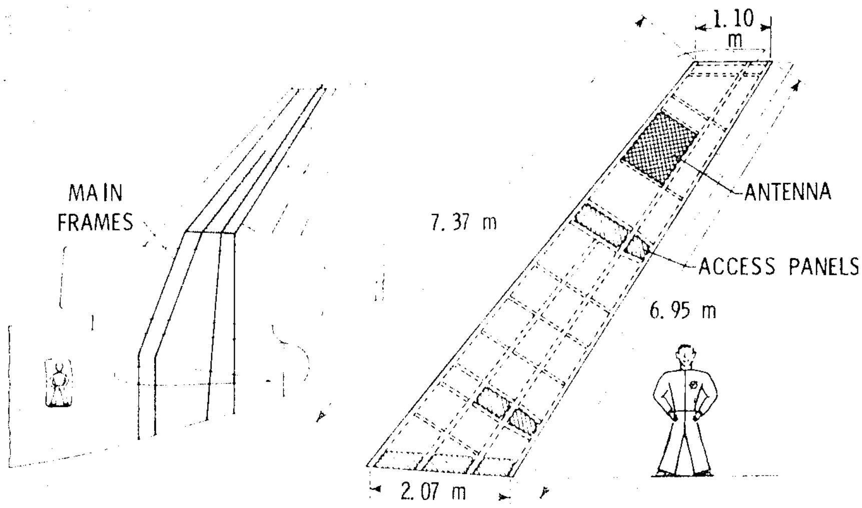


Figure 17.- Overall size and arrangement of DC-10 vertical tail.

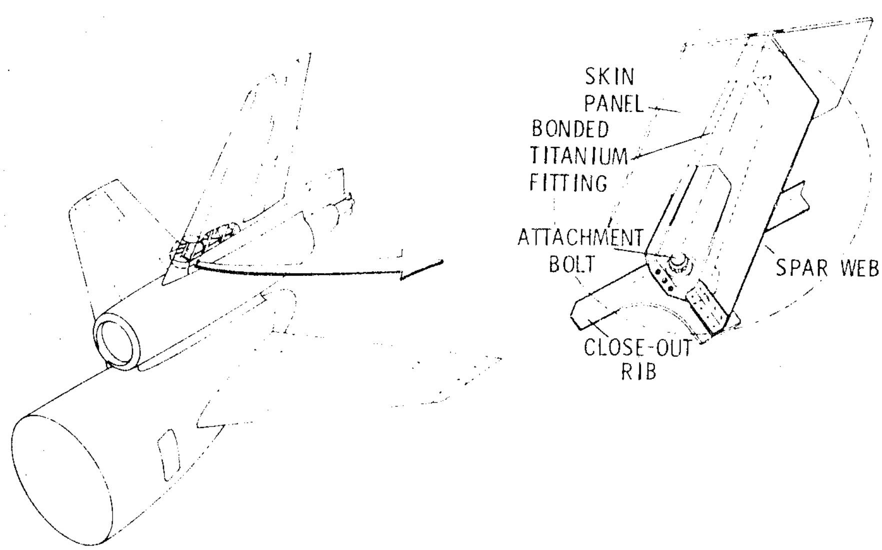


Figure 18.- Titanium spar-root attachment fitting for DC-10 vertical tail.

The present design of the interior structure includes sine-wave webs on both ribs and spars. Honeycomb webs may be used near the root end of the spars to effect the transition to the bonded titanium attach fittings.

A large number of development and design verification test components will be built and tested during the program. Some of the major design verification components are shown on figure 19.

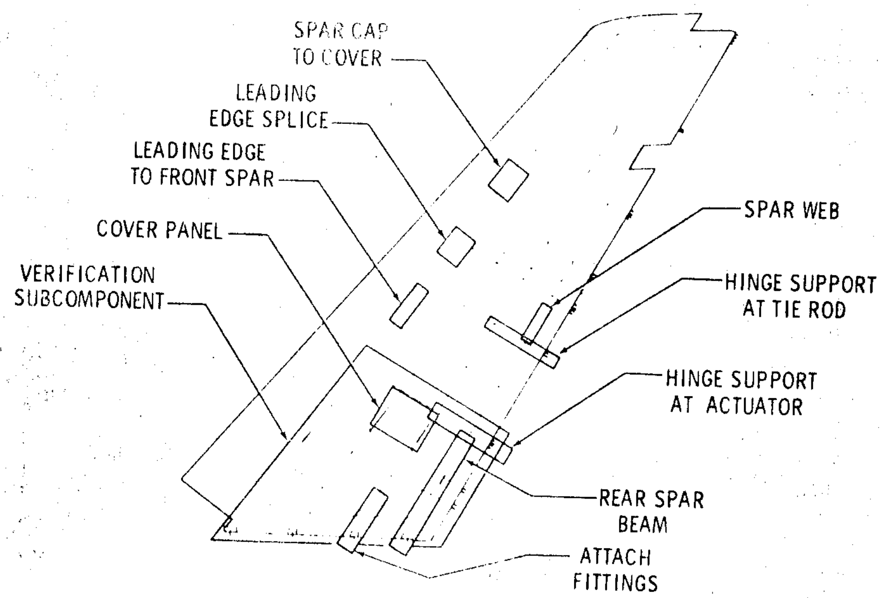


Figure 19.- Major design verification test components for DC-10 vertical tail.

The interior arrangement of the structure is constrained by the location of rudder hinges and actuators, the actuator and root-end attachment access panels, and the antenna. (See figure 17.) However, the improved stiffness characteristics of the composite material are expected to result in about a 27 percent weight savings in the redesigned structure. A preliminary weight comparison is shown in Table VI.

Six vertical stabilizers will be manufactured to verify the processes and obtain manufacturing costs.

TABLE VI.- COMPARISON OF WEIGHT DISTRIBUTIONS FOR ALUMINUM AND COMPOSITE DC-10 VERTICAL TAIL

	ALUMINUM DESIGN WT. kg	COMPOSITE DESIGN WT. kg	WEIGHT CHANGE kg	PERCENT CHANGE
SPAR CAPS	158.4	107.3	- 51.1	-32
SPAR WEBS	62.4	50.2	- 12.2	-20
RIBS	67.9	58.4	- 9.5	-14
COVERS	87.5	61.7	- 25.7	-29
ACCESS DOORS	18.5	16.6	- 1.9	-10
MISC. STRUCTURES	28.7	15.4	- 13.3	-46
TOTAL	423.4	309.6	-113.8	-26.8

SPECIAL DESIGN CONSIDERATIONS

Some of the features of composite structure design are common to all the components being developed. Where possible, the structure is being designed to reduce the number of parts that go into the final assembly. The cocuring of stiffeners and attachment angles to interior ribs and spars reduces the number of parts as well as reducing the number of fasteners used. Wherever possible, skin panels are made as one part with integral, cocured, stiffening. The advantages of one-step curing processes has resulted in some recent changes to the prepreg materials used in graphite-epoxy composites. Most prepreg fabrics and tapes have resin contents in excess of that desired for optimum laminate properties. This excess resin improves the handling qualities of the material during layup but, in most applications, requires a separate pressure-temperature cycle to bleed off the excess resin. Material suppliers have cooperated with the aircraft builders to develop prepregs that do not require bleeding. Although the resin content is still slightly higher than that needed in the final laminate, the new materials are an acceptable compromise between manufacturing ease and laminate properties. Another desirable feature, but one that has not yet been developed, is a self adhesive resin that could be particularly useful for sandwich structure. Self adhesive resins are available for low temperature applications, but the present 450K curing systems require a separate adhesive layer to bond to other materials or to pre-cured laminates.

Because advanced composite materials are more expensive than aluminum, manufacturing cost reductions must be achieved through a reduction in the labor hours that go into each part. In general, fewer separate parts and larger individual parts require less labor than many small parts

Composite designers must provide for adequate corrosion protection. Graphite acts as a noble metal and metals such as steel and aluminum will corrode when in contact with graphite and a suitable electrolyte. When aluminum must be attached to the graphite, a barrier layer, such as glass/epoxy, or a faying surface seal must be used. Also, fasteners must be installed with nonreacting sealants. Generally titanium fasteners are used because titanium is less reactive than aluminum.

The conductivity of graphite-epoxy composites is much lower than that of aluminum and this difference must be considered in relation to the electromagnetic environment of the part. For lightning protection, a conductive layer or conductive straps must be added to the composite part. The proposed lightning protection system for the 727 elevator is shown in figure 20. The system consists of a grid of aluminum straps electrically connected to the mass balance weight, the stabilizer structure and the static discharge probes along the trailing edge. Because of the location on the airplane, only the outboard portion requires lightning protection. The L-1011 and DC-10 vertical fins will have a conductive coating over their entire surface to serve as a ground plane for antennas and to protect the structure from a swept stroke. The metal leading edge and tip structures are most probable points for a direct strike to attach.

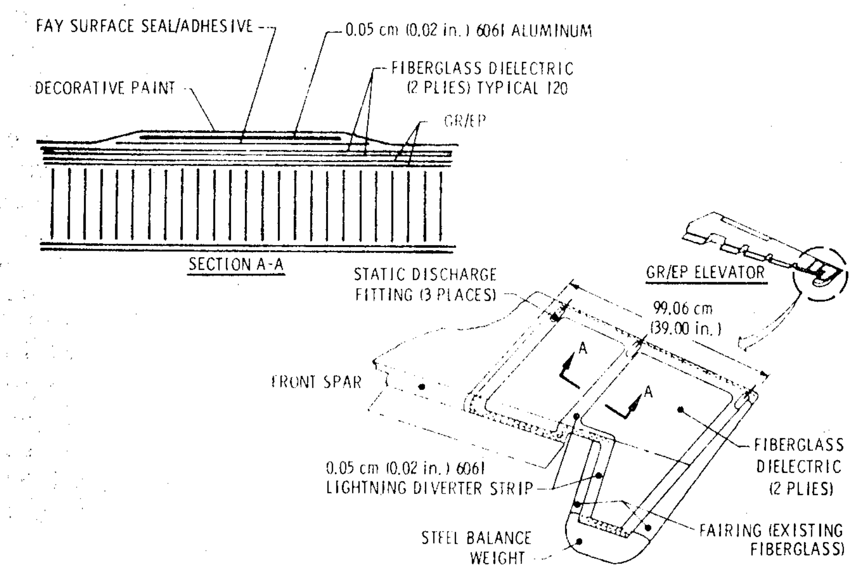


Figure 20.- Lightning protection system for 727 elevator.

CONCLUDING REMARKS

The National Aeronautics and Space Administration's Aircraft Energy Efficiency Program is a cooperative government-industry program aimed toward improving the efficiency of commercial transport aircraft. Emerging technologies, such as advanced composite structures, are being brought to maturity through a series of development contracts. These contracts focus on the design and development of specific structural components to establish the weight savings that can be achieved with production parts in a cost-effective production environment.

The principal U.S. transport manufacturers, Boeing, Douglas, and Lockheed, are each developing two components that have significant weight-saving potential. The components will be totally interchangeable with the existing structures and, if economically practical, could be integrated into current aircraft production. In addition, the components will serve as prototypes for the development of similar classes of composite structures on future new aircraft.

The components are in various stages of development at this time. Preliminary weight estimates show from 25 to 30 percent weight savings from the use of composites. Design concepts and methods of construction are being selected that will enhance structural reliability and keep manufacturing costs low. For all the components presently under development, the goal is to make the composite component cost less than its metal counterpart.

As aircraft builders gain experience in producing composite hardware and develop confidence in their designs and manufacturing cost projections, the use of composites will increase substantially.

REFERENCES

1. Povinelli, Federick P.; Klineberg, John M.; and Kramer, James J.: "Improving Aircraft Energy Efficiency," Aeronautics and Astronautics, Vol. 14, No. 2, February 1976, pp. 18-31.
2. Lehman, George M.: Development of an Advanced Composite Rudder for Flight Service on the DC-10. Third Conference on Fibrous Composites in Flight Vehicle Design. NASA TM X-3377, pp. 71-91. November 1975.
3. Lehman, George M.: Advanced Composite Rudders for DC-10 Aircraft - Design, Manufacturing, and Ground Tests. NASA CR-145068, 1976.
4. James, A. M.; Fogg, L. D.; and VanHamersveld, J. A.: Design of Advanced Composite Ailerons on Transport Aircraft; Summary Report. NASA CR-132637, 1975.
5. James, A. M.; and Fogg, L. D.: Design of Advanced Composite Ailerons on Transport Aircraft; Design Report. NASA CR-132638, 1975.
6. Stauffer, Warren A.; and James, Arthur M.: Development of Advanced Composite Structures for Lockheed Aircraft. CTOL Transport Technology - 1978. NASA SP-2036, Part I, pp. 259-280.